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Structural Integrity and Aging-Related Issues for Helicopters

Joyanto K. Sen
MS B530-B346
The Boeing Company
5000 E. McDowell Road
Mesa, Arizona 85215, USA

and

Richard A. Everett
U.S. Army VTD
MS 188E
NASA – Langley Research Center
Hampton, Virginia 23681, USA

INTRODUCTION

The question of the structural integrity of aging aircraft became an issue of grave concern when an Aloha Airlines Boeing 737 suffered major structural damage in April 1988 while in flight. Since then the airworthiness issue of aging aircraft has been the concern of manufacturers, and civilian and military operators alike. The issues for civilian and military operators are structural integrity and reduced ownership cost when the service life is extended. The military have the additional task of maintaining preparedness with improved availability and enhancing the performance of aircraft designed for now-obsolescent missions to meet new mission requirements. Aging, therefore, does not mean "old" in terms of the number of calendar days, but the cumulative effect of technical obsolescence, changing requirements, quality of maintenance and the nature of operation (i.e., load and environment).

The issues of structural integrity for rotary-wing aircraft are somewhat different from those of fixed-wing aircraft. In helicopters, the dynamic rotor components are safe-life designs and are replaced at the end of their service lives. Thus, airworthiness concerns of structural integrity for helicopters are limited but still pose great challenges in adjusting to changing missions. Structural integrity issues for helicopters are in the airframe, equipment and avionics, and retention hardware for non-airframe related structures. Most of the rotary-wing aircraft in the U.S. Army's inventory are several decades old, and are required to continue in service even longer, Table 1 (Ref. 1). They were designed for missions that have changed and with equipment that have been overtaken by technological advances. Thus, the primary issues for aging military helicopters have been to assure structural integrity while enhancing performance with more capable dynamic components and technically advanced equipment.

The average age of aircraft in the U.S. Navy's inventory show a similar trend for rotary- and fixed-wing aircraft, Fig. 1 (Ref. 2). The average age of helicopters is 19.2 years and continues to climb until 2005, when aggressive procurement of new helicopters will lower their average age. To continue to field these aircraft, their upgrades should be a planned, continuous improvement process in order to dovetail each stage of upgrade and prevent the cost associated with a one-time upgrade. Because of the severe environment in which the U.S. Navy helicopters operate, it is more convenient to replace life-limited parts than to inspect them to damage tolerance or other aging aircraft requirements in order to assure structural integrity under extended service-life procedures, Ref. 3. Damage tolerance for the U.S. Navy is a "band aid – short-term solution," only to "maintain immediate flight safety, eliminate the problem and return to safe-life operation."

To focus on the structural issues of aging helicopters, the causes of worldwide accidents of civilian helicopters in 1999 are examined in Fig. 2, Ref. 4. The majority of the accidents were due to pilot error followed by engine failure or power loss. Six structural failures were the causes for 3.1 percent of the accidents. The six structural failures were: (1) the tailboom separated on a Bell 407, (2) the retaining nut and bolt for the tail rotor shaft failed on a Bell 206L1, (3) "major mechanical failure" on a Hughes 369D, (4) a main rotor tension-torsion strap failed on a BK 117B-1, (5) a HH-43F was seen to "explode and disintegrate in flight," and (6) bolts in the mounts of the 90° gear box failed. The tension-torsion strap is a safe-life design and is replaced at the end of its service life. The failures of the tailboom, the tail rotor shaft retaining system and the mounts of the 90° gear box are structural issues concerning aging helicopters. The age of these aircraft and the precise causes of structural failure are not known.

Table 1 Age of Rotary-Wing Aircraft in the U. S. Army (Ref. 1)

Type	Number of Aircraft	Average Age, Years	Retirement Date
OH-58 A/C	474	35.0	2017
OH-58D	387	12.5	2024
CH-47D	466	20+	2033
AH-1	389	30+	2017
UH-1H/V	1073	29.0	2025
UH-60A	906	17.0	(Not Set)
UH-60L	515	6.0	(Not Set)

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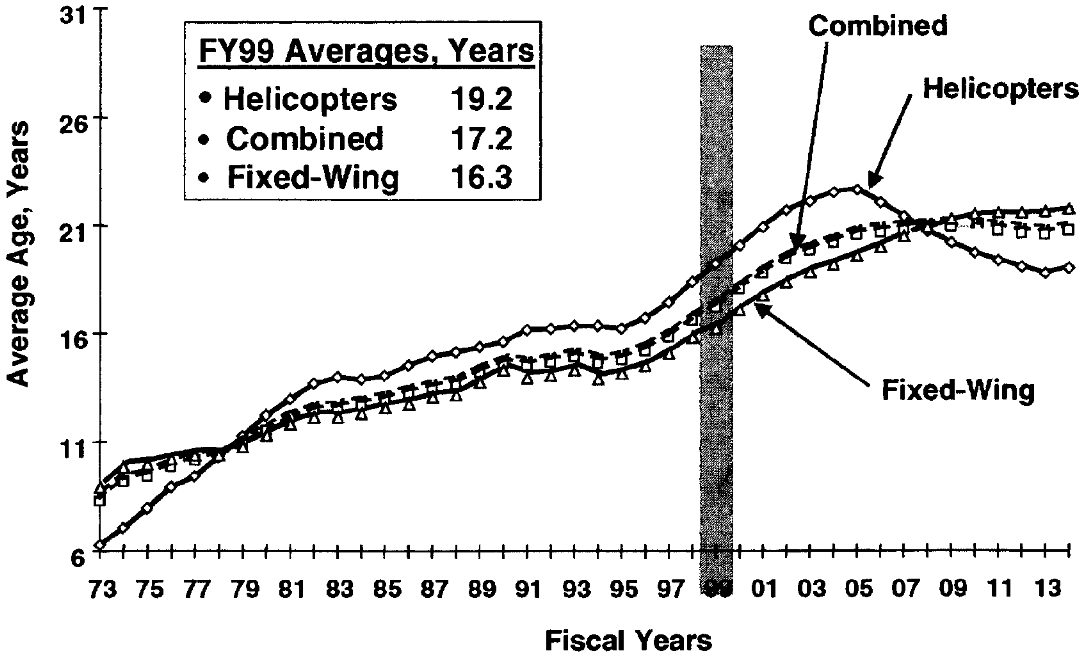


Fig. 1 Average age of U.S. Navy aircraft (Ref. 2)

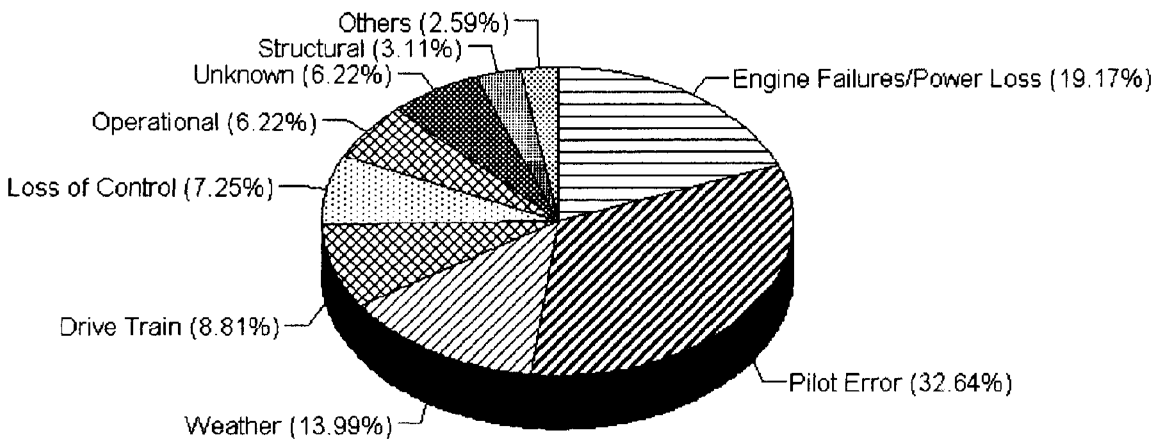


Fig. 2 Causes of worldwide civilian helicopter accidents in 1999 (Ref. 4)

The aging issues among helicopter operators, ordered in terms of the total cost, are engines, dynamic components and drive train, equipment, airframe, and avionics. For medium and heavy helicopters, the breakdown of the acquisition cost and direct maintenance cost (DMC) as percentages of the total cost are shown in Fig. 3, Ref. 5. The total cost is the sum of acquisition cost, DMC and operating cost. The percentages for insurance, fuel and crew – elements of the operating cost – are also shown in the figure.

Ignoring the cost of labor, the DMC for engines, dynamic components and drive train, avionics and equipment are much higher than that for the airframe. The DMC for airframe is lowest at 10 percent of the acquisition cost, while that of dynamic components is

138 percent. The DMC of dynamic rotor components is high because of replacement cost following their useful service life. The engines and drive train are specialized designs and their aging issues will not be discussed here. Engine and drive train designers are contributing to enhancing reliability through simplified designs and higher power-to-weight ratio systems. Avionics will also not be discussed as avionics designers address cost and reliability through the development of a family of avionics systems, and address continuous upgrades with open operating systems. Equipment includes fuel cells, wiring harnesses and retention hardware of all systems. Wiring systems "age" because of environmental conditions, vibration, operational wear and tear, and improper repair.

This overview presents the structural integrity issues in extending the service lives of dynamic components and airframes of aging helicopters. With the use of composites, the acquisition cost and DMC of dynamic components have reduced greatly over the last two decades. This reduction has been possible because innovative designs of complex geometry can be fabricated more accurately and cost-effectively with composite manufacturing technologies. In addition, significant developments have occurred to make rotor design more efficient, Ref. 6. However, the methodologies for calculating service lives and their optimum applicability are still being discussed. These issues on structural integrity are presented and discussed below. Even though all-composite dynamic components have been in service for several decades, an all-composite airframe is still not a reality. The total composite content in the airframe and rotor of production helicopters in 1992 was around 25 percent, Fig. 4. Thus, some of the aging issues of fixed-wing aircraft are still applicable to metal airframes of the helicopter though the problems are less severe.

STRUCTURAL INTEGRITY ISSUES FOR HELICOPTERS

In the design of rotorcraft structures, the objective of assuring structural integrity is to reduce to zero the probability of catastrophic failure. Rotorcraft structures can be classified as two distinct types: dynamic components and non-dynamic components of the airframe. Dynamic components are those of the rotor systems and are subjected to high-cycle oscillatory loads. As shown in Fig. 2, loss of structural integrity was the cause of 3.1 percent of the 1999 accidents surveyed. To get a better understanding of the typical causes of, and the frequency of occurrence of these causes in, rotorcraft accidents, an Eurocopter study of accidents over five years on a worldwide, all-missions basis identified 37 accidents per one million flight hours, Ref. 7. Of the 37 accidents, 16 resulted in fatalities. The precise causes of structural failures were identified and were found to be responsible for only 0.3 percent of the accidents, while incorrect maintenance was responsible for 17 percent and engine malfunction for 3.1 percent, Fig. 5. The predominant cause, constituting 77 percent of all accidents, was operational and environmental conditions. The 0.3 percent from structural failures were due to "poor design, non-conformity of components, more severe load spectrum than expected" and non-identified causes of fatigue cracks.

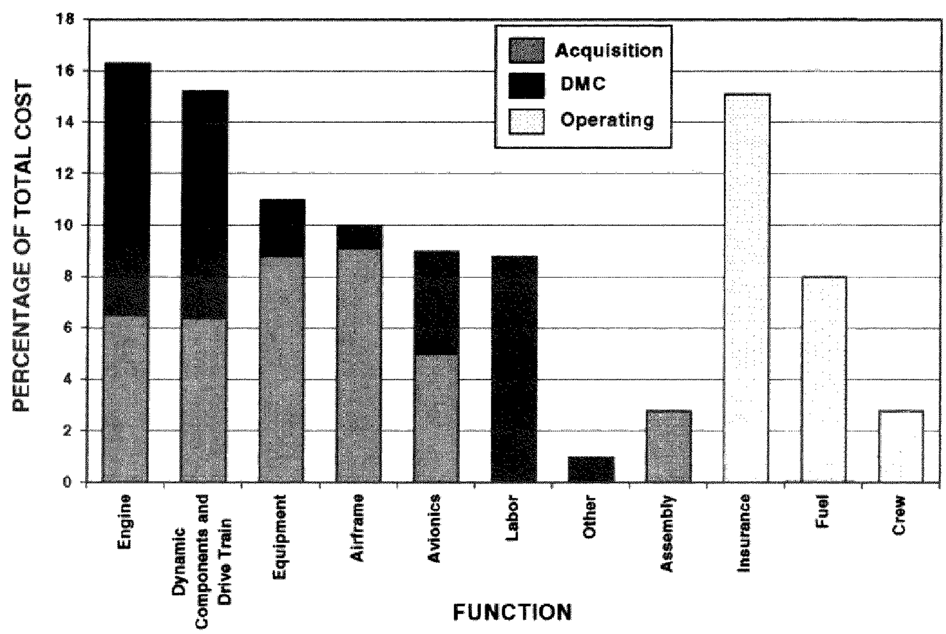


Fig. 3 Breakdown of costs for medium and heavy helicopters (Ref. 5)

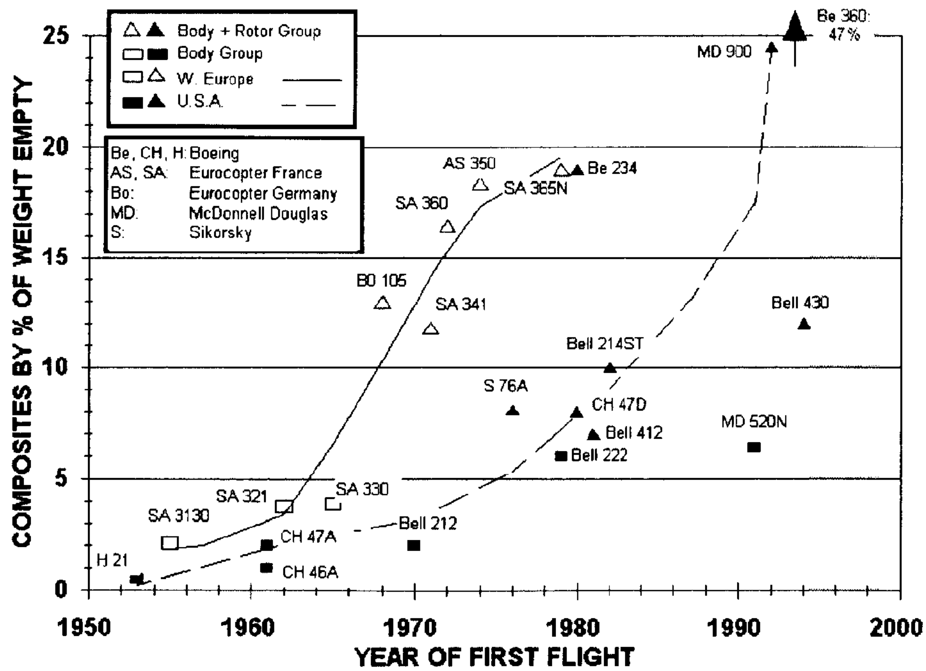


Fig. 4 Application of composites in helicopter to 1992 (Ref. 6.)

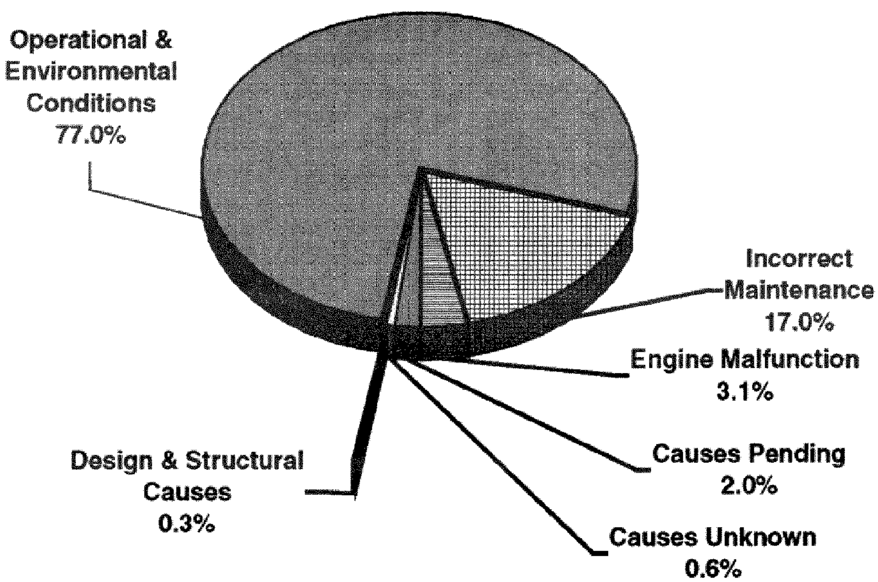


Fig. 5 Causes of 37 rotorcraft accidents per a million flight hours (Ref. 7)

The causes of, or catalysts for, fatigue cracks leading to failures can be assigned to the development, manufacturing and operational processes of rotorcraft.

In development, causes include

- design oversights,
- underestimating loads, and
- analyses oversights.

In manufacturing, causes or catalysts include

- planning oversights
- non-conformal parts, and
- manufacturing defects.

In operation, causes or catalysts include

- usage more severe than expected,
- corrosion,
- improper or inadequate service, and
- service defects.

The defects introduced during manufacturing or in service are not causes but act as catalysts that initiate fatigue cracks, which may then lead to failures. The survey of helicopter accidents in Reference 7 identified catalysts for the cracks, and the frequency with which they occur is shown in Fig. 6.

Rotorcraft manufacturers validate structural integrity and assure, through test and analysis, that the level of baseline integrity will be maintained during operation. The manufacturers also provide guidance on how this integrity can be continued to be assured in service through inspection, and what provisions are available when the integrity is sufficiently degraded where safety of flight will be jeopardized.

Of all the structures fabricated by helicopter manufacturers, dynamic components are the key to the helicopter's performance to stated requirements. Dynamic components also have the highest DMC compared to all other components, Fig. 3. Dynamic components are subjected to large numbers of spectra of oscillating loads and generally fail in fatigue. Several approaches are taken by manufacturers to certify dynamic components. The terminology used to describe these approaches will be first defined, followed by descriptions of, and discussions on, the substantiation methodologies. The aging-related issues and the differences between certifying metallic and composite components are included in the discussions. The merits of the various definitions of the terminology are not discussed and the primacy of any one methodology is not championed.

There are four "traditional" approaches for substantiating the life of components in order to assure safety of flight: safe-life, damage tolerance, fail-safe and flaw tolerance. These four approaches form the basis of the decision process on whether a part be retired and replaced based on accumulated flight hours, or retired based on the damage sustained, or returned to service after repairing the damage. From practical considerations, components may be substantiated by any one of the approaches or by a combination of several approaches. The definitions of the four traditional approaches are given below.

Definitions

Safe-Life

Safe-Life is defined by the US Federal Aviation Administration in Reference 8 as,

> "Safe-Life of a structure is that number of events such as flights, landings or flight hours, during which there is a low probability that the strength will degrade below its design ultimate value due to fatigue cracking."

The safe-life approach assigns a finite life to a component. This definition focuses on the baseline strength and its degradation in operation. The definition also infers, and its application alleviates, the difficulties and cost of inspecting complex rotorcraft structures. The static and fatigue strengths and their progressive losses can be determined through test, or in combination with analysis. The safe-life approach is based on the remote possibility of a crack initiating in a component, and it recommends that the component be retired when accumulated flight hours have completed the assigned finite life or when a crack is detected by currently available means.

Composite components are never as pristine as metal components. In contrast to metals, the influence of detectable damage in composites is difficult to assess because composites often exhibit "cosmetic" damage, which should not be the basis for limiting the structural life or be the reason for redesign. Thus, for a composite part to be retired, the damage must exhibit structurally significant delamination, splintering, matrix cracking and fiber breakage.

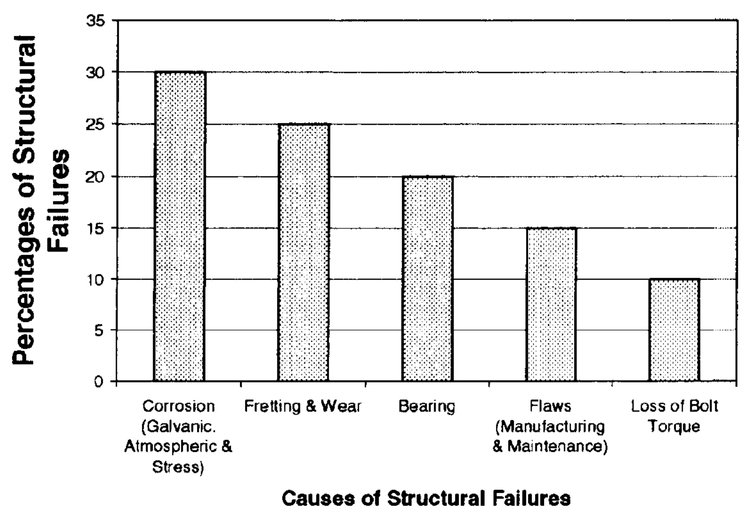


Fig. 6 Frequency of the causes of the structural failures of Fig. 5 (Ref. 7)

Damage Tolerance

Damage tolerance is defined by the U.S. Air Force in Reference 9 as,

> "Damage tolerance is the attribute of a structure that permits it to retain its required residual strength for a period of unrepaired usage after the structure has sustained described levels of fatigue, corrosion, and accidental or discrete source damage such as (a) unstable propagation of fatigue cracks, (b) unstable propagation of initial or service-induced damage, and/or (c) impact damage from a discrete source."

The focus in this definition of damage tolerance is to quantify the level of damage that a structure can tolerate and retire or repair it before a catastrophic failure occurs. It assumes that any structure is essentially imperfect as a result of the "inherent material structure, material processing, component design, manufacturing or usage," Ref. 10. In order to quantify the level of tolerable damage, the damage must be assessed, and the rate at which this damage will propagate and the damage level at which the residual strength will fail to react the loads must be calculated. The principles of fracture mechanics are used to evaluate the damage tolerance of a structure, and to calculate the period and level of inspection required to mitigate the risk of failure.

The elements of the procedure for calculating the time for crack initiation and the rate of crack propagation for metals are applicable to composites. The difference, however, is in inspection. Inspection is a subjective process, and assessing "damage" in composites is more than measuring the length of a crack. Therefore, training should maximize on the inspector's experience, the type of structure and material in order that the damage criterion for composites is applied uniformly.

Fail-Safe

Fail-Safe as defined in Reference 9 states

> "Fail-safe is that attribute of the structure that permits it to retain its required residual strength for a period of unrepaired usage after the failure or partial failure of a Principal Structural Element (PSE)."

A PSE is an element of the structure whose integrity is essential for maintaining the overall structural integrity of an aircraft. Even though the fail-safe concept states that residual strength is essential to achieve redundancy, the concept does not develop inspection requirements to monitor damage. Fail-safe designs, therefore, provide multiple load paths with redundant structures such that the failure of one load path will safely distribute the applied loads to other load-carrying members.

Flaw Tolerance

The flaw tolerance approach is based on the premise that flaws exist in any structure, and act as catalysts for initiating cracks. The flaw tolerance approach advises inspection and recommends that a component be retired after the prescribed flight hours are accumulated or when a crack is found in inspection. This approach does not recommend periodic inspections to monitor crack growth or subscribe to any residual strength requirements. However, it does require the maximum tolerable flaw sizes in critical locations be determined based on historical data. The data on the size and the location of flaws are then used to conduct constant-amplitude fatigue tests of flawed specimens. Replacement times are then calculated using the safe-life approach. This approach is also known as "enhanced safe-life," and is "sometimes used in combination with or in place of" flaw tolerance, Ref. 11.

SUBSTANTIATION METHODOLOGIES FOR DYNAMIC COMPONENTS

All components subjected to oscillatory loads above the endurance limit will accumulate damage, which result in cracking and wear, and inevitable failure. The helicopter manufacturers have tended towards two methodologies depending on the material of the component, its design configuration, its load spectrum and how critical is its function in the helicopter's operation. The two methodologies are safe-life and "extended safe-life." "Extended safe-life" is a new terminology, and is used in this context to represent a combination of flaw tolerance and damage tolerance. Included in the extended safe-life approach are features of the fail-safe approach where multiple load path designs are featured in critical structures. The features of the four traditional approaches that constitute the two methodologies are summarized and compared in Table 2. The extended safe-life methodology was accepted as viable by representatives of several U.S. helicopter manufacturers represented on the Fatigue Methodology Committee of the Rotorcraft Advisory Group under the Civil Aviation Council of the Aerospace Industries Association, Ref. 12. The improved substantiation process proposed by Eurocopter, Ref. 7, encompasses elements of the extended safe-life methodology described here.

Safe-Life Methodology

Traditional fatigue tests on full-scale components are conducted to characterize their fatigue behavior in the form of S-N curves. The tests are conducted on as-manufactured parts and are subjected to constant amplitude loads based on measured flight loads. The service life is computed with a high safety factor based on the component fatigue strength. The high safety factor applied to as-manufactured parts has to-date accommodated the strength-reducing flaws in structures with a high degree of reliability as shown in Figures 2 and 5. The safe-life methodology, therefore, assumes that a structure has a finite fatigue life which can be estimated from experimental results and analysis. The finite life, or the safe service life, determines the

Table 2 Summary of Substantiation Methodologies

Feature	Safe-Life	Extended Safe-Life: Damage Tolerance	Extended Safe-Life: Fail-Safe	Extended Safe-Life: Flaw Tolerance
Focus	• Baseline strength • Strength degrades in service	• Level of tolerable damage quantified	• Redundant or multiple load paths	• Maximum flaw size in critical locations
Retirement Criterion	• Prescribed flight hours accumulated • Crack detected	• On-Condition, i.e., damage degrades residual strength below acceptable level	• Residual strength below acceptable level when a redundant load path fails	• Prescribed flight hours accumulated • Crack detected
Structure	• Assumed pristine	• Assumed flawed	• Assumed pristine	• Assumed flawed
Periodic Inspection to Assess Criticality of Crack	• No	• Yes	• No	• No
Substantiation Methodology	• Traditional fatigue	• Fracture mechanics	• Traditional fatigue	• Traditional fatigue based on flawed specimens

accumulated flight hours allowed before a part is replaced or retired.

The basic elements of the safe-life methodology for metals are constant-amplitude fatigue tests with accelerated loading to develop the component S-N curve, standard S-N curve shapes, usage spectrum developed from measured flight loads, standard safety reduction factors, and Palmgren-Miner's Rule of linear cumulative fatigue damage. In the case of composites where the S-N shape curve is typically flatter than for metals, load excursions, the cycles to crack initiation and the Palmgren-Miner's Rule can greatly influence the component life calculated from the safe-life methodology.

Composite designs are generally damage tolerant. However, because the S-N shape curves of composites are flatter than of metals, they are sensitive to load excursions. A relatively small increase in load results in a large decrease in the allowable number of cycles on the S-N curve. If this load increase is associated with steady-state flight regimes or with frequently occurring maneuvers, the calculated retirement life based on the safe-life methodology could be dramatically reduced.

The safe-life methodology requires a component be retired when a crack is initiated. In composite structures matrix cracking is typically the damage mechanism that leads to delamination. It is difficult to determine when the fatigue strength of a composite structure has significantly degraded. Since the number of cycles to crack initiation is plotted to estimate the service life, an error in plotting it on a comparatively flat S-N curve can result in a greatly erroneous service life.

Since the S-N curve is flatter in composite structures, damage is generally due to a few high loads in the spectrum, such as ground-air-ground loads. The damage may result in several matrix cracks and delaminations. The number of cycles to failure could be the cumulative effect of their initiation and propagation. Since the load representing the cumulative effect is difficult to quantify, Palmgren-Miner's rule must be cautiously applied.

Extended Safe-Life

The extended safe-life methodology uses a combination of flaw tolerance and damage tolerance approaches. Component replacement times are calculated using the traditional safe-life approach with full-scale fatigue tests on specimens with flaws representative of defects that occur in manufacturing and in service. However, in addition to establishing failure, the tests monitor crack growth to failure. Damage tolerance principles then establish inspection periods for cracks and to evaluate degradations in fatigue or quasi-static strengths. By this combination of approaches, parts can not only be retired when loss of function occurs but component service life can be increased when validated by inspection.

The flaws intentionally inflicted on specimens are the maximum probable flaw sizes determined from historical data from manufacture and service. In order to establish inspection procedures, damaged areas must be accessible during service, and cracks must be detectable and measurable by the method and tools that will be used to perform this inspection. The accurate determination of the initial damage size and its propagating length must take into account the experience and training of the personnel who will be performing the inspection. This

approach, therefore, promotes user-friendly designs for operators and maintenance institutions.

In order for this methodology to provide even higher reliability and to significantly reduce accidents, components and sub-assemblies can be designed with multiple load paths and to provide easy access for inspection in critical locations where tests have indicated that damage will occur. As a standard procedure, the extended safe-life approach establishes replacement or retirement lives of principal structural components. Components are designed to be tolerant to flaws, to propagate cracks slowly, and to provide redundant load paths in critical structures. These features, together with planned inspections to assertain that crack sizes are below acceptable limits for components to perform their stated functions, will assure that components will successfully react the spectrum of operating loads until the next inspection period. This approach is applicable to metals and composites.

AIRFRAME SUBSTANTIATION PROCEDURES

Metal and composite airframes are designed to meet crashworthiness and ballistic-tolerance requirements. These static requirements establish the static design criteria where the static limit loads exceed the operational oscillatory loads by a large margin. Rotor-generated oscillatory loads in helicopter airframes are also significantly below the loads from these static requirements. Low-cycle airframe fatigue loads from normal landings and maneuvers, although higher than rotor-generated oscillatory loads, are still well below the static design criteria.

Metallic structures, with high stress concentration areas, generally have low fatigue endurance limits compared to their static ultimate strengths. Composites, on the other hand, with flatter fatigue S-N curve shapes and low fatigue sensitivity to stress concentrations, tend to have relatively high fatigue endurance limits in tension-dominated modes compared to their static ultimate strengths. Experience has shown that a composite airframe with good static strength will have high fatigue strength margins. This means that a fatigue test of a full-scale, composite airframe may not be necessary provided that analysis, based on a building block approach, validates that the oscillatory loads do not exceed the endurance limits.

Full-scale fatigue tests of the airframe must be conducted when detail analysis cannot be corroborated by test in highly loaded areas or when the load path is complex. In these cases, the test may be on the full airframe or a major sub-assembly of the airframe. Occurrences of a large number of high oscillatory loads, large out-of-plane loads, highly loaded complex joint configurations in major bulkheads or when the effects of loads are not known are all reasons for conducting full-scale airframe tests. Additionally, any airframe designed for repeated heavy lift missions are candidates for full-scale airframe fatigue tests.

In the manufacture of the helicopter airframe, a large number of flaws may be permitted in order to reduce the costs of production, inspection, and rework. Since composites are inherently damage tolerant, any damage-tolerant features in airframe design only enhances the fail-safety of composite construction. A "no-growth" qualification on this basis requires a low strain level that further reduces the possibility of generating fatigue damage or propagating flaws.

The presence of a crack or a flaw in an airframe structure does not preclude it from being airworthy. However, in order to demonstrate airworthiness, a full-scale test under representative loads may be necessary. Cracks in metal structures or delaminations in composite structures may be acceptable if the flaw growth rate or the rate at which new cracks appear in adjacent structures are deemed inconsequential to the overall structural integrity.

As an example, the fatigue test on the all-composite tailboom for Boeing's MD 500N helicopter produced two unexpected benefits. Very early in the tests, cracks developed in an area subjected to high out-of-plane bending loads. The locations of the cracks confirmed the high strains predicted by finite element analysis. The growth of the cracks was monitored and found to be arrested after an additional 225 simulated hours of flight. No further growth was measured after 4,100 simulated flight hours and two applications of enhanced limit load. The test validated the crack to be benign and detectable, and was used to establish a safe inspection interval, Ref. 12. A field repair was then designed and tested on the same test article for the duration of the fatigue test. The repair procedure was qualified for field application and inspection procedures established.

A damage tolerance approach to the design of helicopter airframes is almost always chosen, and composite construction makes this an even more advantageous choice. The full-scale fatigue test must take into account the material and operational variability while demonstrating structural integrity. The variability may be demonstrated by (1) a test conducted under environmental conditions; (2) applying a scatter factor to the fatigue test time; (3) multiplying the test loads by a load enhancement factor (LEF); or (4) a combination of all three.

The full-scale fatigue test for Boeing's MD 500N tailboom was conducted under hot-wet conditions using an LEF of 1.18 and a scatter factor of 2 on life. The tailboom was soaked for 30 days at 85 percent relative humidity and a temperature of 71°C before the test. If a lower load enhancement factor were selected, the test duration would have been longer. If the failure mode is unknown or there is more than one failure mode, the highest applicable LEF must be used in order that all possible failure modes are considered. However, the selected LEF must not be so large where high deflections

result in false failure modes. In tests of metal airframes, LEFs are generally not used. When testing a composite or a composite-cum-metal airframe, a high LEF must be carefully chosen in order to avoid qualifying an over-designed structure or recommending early retirement or repairing the airframe during the test. The airframe must be analyzed in detail for overloads and high local residual stresses in order to select the appropriate LEF. This methodology is based on that developed for fixed-wing aircraft, Ref. 13.

The size of the airframe for the full-scale test may make moisture and temperature conditioning impractical. A combined test and analysis procedure is then used. The full-scale static test article is instrumented at critical locations where the measured strain can correlate the finite element analysis. Once the correlation between the test and the finite element model has been established, the maximum strains for all the critical loading conditions can be calculated. The maximum strains are then compared to the material allowable data to validate the structural integrity of the airframe design.

AGING AIRCRAFT ISSUES FOR FIXED-WING AIRCRAFT AND HELICOPTERS

As stated in the introduction, aging aircraft concerns were brought to the public's attention when the Aloha accident occurred in 1988. This type of problem could be expected in the civilian fixed-wing fleet because of the number of aircraft operating and the many hours that these aircraft fly. Although structural aging problems could be thought of as those relating to years in service, they can also be attributed to the cumulative effect of cyclic loading during operation. From the context of structural integrity, the fixed-wing fleet has identified corrosion and wide-spread fatigue damage (or, multi-site damage) as the two primary aging aircraft issues. The civilian fleet has an advantage over the military fleet in that replacement of older aircraft is more likely than in the military fleet where, since the end of the cold war, the military has continued to cut back their acquisitions, and aircraft are now expected to remain in service for as long as 50 years (see Table 1).

Although the FAA addressed the aging problem issues as early as 1968, Ref. 14, with new procedures issued in 1978 for maintaining the safety of aircraft as they age, it was not until the Aloha accident that a national program was developed to study in detail the problems associated with aging. Along with several conferences that focused on aging issues and several prominent research programs that have come into existence, an Aging Aircraft Task Force (AATF) was formed. An independent advisory panel, Technical Oversight Group for Aging Aircraft (TOGAA), was appointed to continuously review the aircraft industry and airlines. TOGAA began their review of the helicopter civilian fleet in 1994. At one of the industry and FAA review meetings in 1995 the helicopter industry stated that "aging is not a significant issue for rotating parts because they are replaced or extensively refurbished on a periodic basis" as a result of the safe-life design philosophy. Even though TOGAA agreed with this statement, the helicopter community would consequently state that issues such as corrosion and multi-site damage are fatigue phenomena that do occur in helicopters.

Corrosion is a problem that obviously occurs on most metal structures that operate in a salt or moisture environment. As to the second primary aging phenomenon of multi-site damage, known as MSD in the fixed-wing community, different experiences seem to exist depending on the operation of the helicopter. If MSD does occur it is often more an economic concern for helicopters in repairing these multiple damages rather than a safety issue. Quite the reverse is the case with fixed-wing aircraft. While there are some obvious differences in aging problems between helicopter and fixed-wing aircraft structures, there are also some similarities. These differences and similarities on corrosion, MSD, structural inspections, loads monitoring, and the separate problems associated with military and civilian operator are discussed below.

Corrosion

It is obvious that corrosion is a problem for both the fixed- and rotary-wing aircraft since both are still predominately metal structures. Since many aircraft operate around a salt water environment (U.S. Navy, U.S. Coast Guard, and helicopter operators of off-shore oil platforms in the North Sea and the U.S. Gulf Coast), the corrosion problem requires constant vigilance in both preventing corrosion and repairing corrosion damage. Currently no mathematical model exists that can predict the rate of accumulation of corrosion. In fact because of the many variables that effect the accumulation of corrosion (mostly where the rotorcraft is being used), it is probably not possible to predict the accumulation of corrosion without a usage monitoring system such as a Health and Usage Monitoring System (HUMS) unit. With a usage monitoring system capable of tracking corrosion as it accumulates, an on-board computer could predict the life of a fatigue crack propagating in the corrosive environment. The technology to predict fatigue crack growth in a known corrosive environment exists today.

As is the case with the fixed-wing community, a computer database that reflects all of the experiences of rotorcraft in a corrosive environment is not available today. The U.S. Coast Guard has developed a limited database two years ago. In fixed-wing military operations where the U.S. Air Force has been tracking structural problems through their Aircraft Structural Integrity Program, ASIP, since the early 70's, a database on corrosion problems has been recorded prior to 1990 for only the KC-135 aircraft. The KC-135 has been in operation for 40 years, and there is no obvious date for it to end service, which is another example of how military

aircraft will continue to age. Since the military had not planned on keeping aircraft in service for such extensive periods, aging considerations were not considered in their original designs. Also, many of these aircraft were designed in the 1950's with materials that are more susceptible to corrosion than currently available materials. This problem is illustrated in the case of a main rotor grip on an "older" helicopter that was made of 2014-T6 aluminum alloy, Fig. 7. After corrosion problems occurred, the material was changed to 7075-T73 aluminum alloy which offers greater resistance to stress corrosion, Ref. 15. One design concept that occur in all airframe structures that lead to corrosion is the lap splice joint in fuselage skin construction. The area of the lap joint has been shown to initiate and accumulate corrosion. The lap splice joint problem can be selectively, though not completely, eliminated in helicopter airframes. The joint is more extensively used in fixed-wing aircraft.

One weakness that can cause problems in helicopters that does not occur in most fixed-wing aircraft concerning corrosion is the safe-life design methodology. Traditional safe-life does not account for any deviation in fatigue strength that may occur over time (aging) due to flaws or corrosion that develop during manufacturing, maintenance or service. However, in 1988 for helicopters the flaw tolerance design methodology was added to the Federal Air Regulation to help alleviate this short-coming of the traditional safe-life design method. Several approaches can be taken to modify the safe-life methodology after the helicopter has entered service. These are illustrated below.

In the case of the structural life management of a horizontal hinge pin for the CH-53 A/D helicopter, which had been an out-of-production U.S. Navy helicopter, a study of the conditions that can reduce structural integrity was undertaken in order to extend the life of this component, Ref. 16. The horizontal hinge pin is made of 4340 steel and had experienced corrosion problems during its service life. Using flaw tolerance concepts a coupon test program was developed using a "worst case" corrosion pit. The coupon specimens were tested in fatigue and showed a 63% reduction in fatigue strength, Fig. 8. This study set the inspection interval at 1,200 hours, which coincided with a scheduled overhaul and which allowed regular inspections up to its normal retirement time of 8,300 hours.

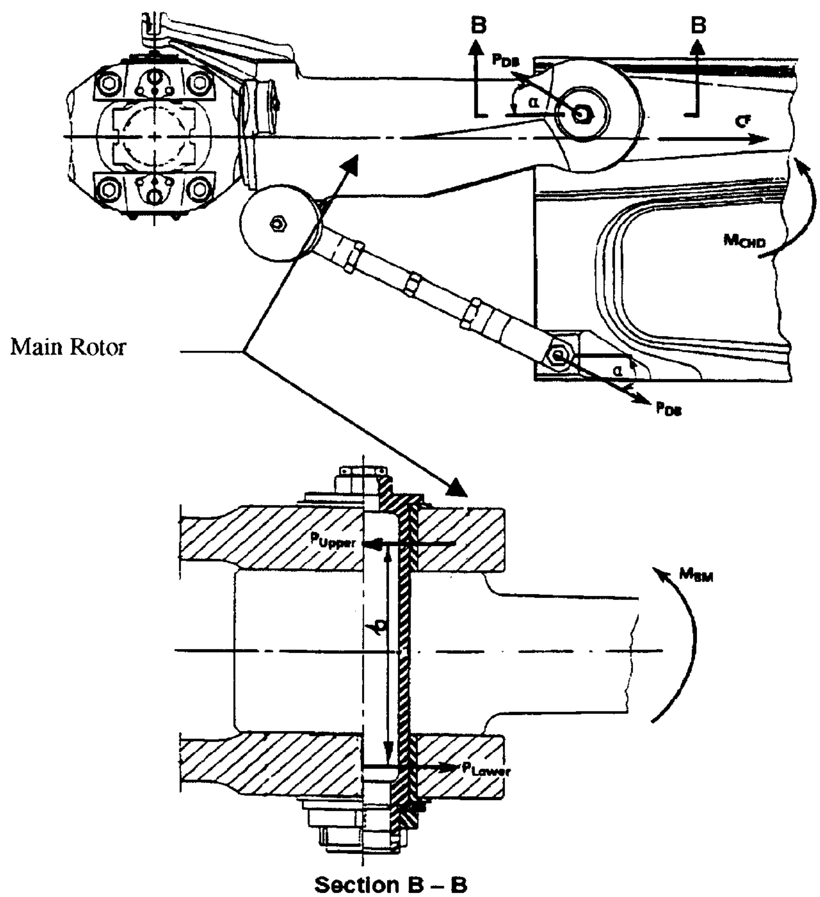


Fig. 7 Main rotor grip where corrosion was mitigated by replacing the aluminum alloy (Ref. 15)

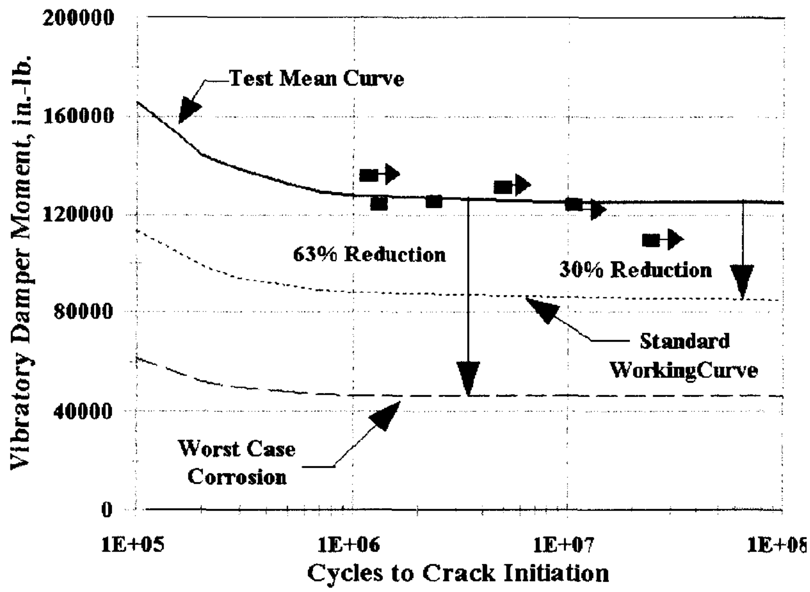


Fig. 8 Effect of corrosion on the S-N curve of the CH-53 A/D horizontal hinge pin (Ref. 16)

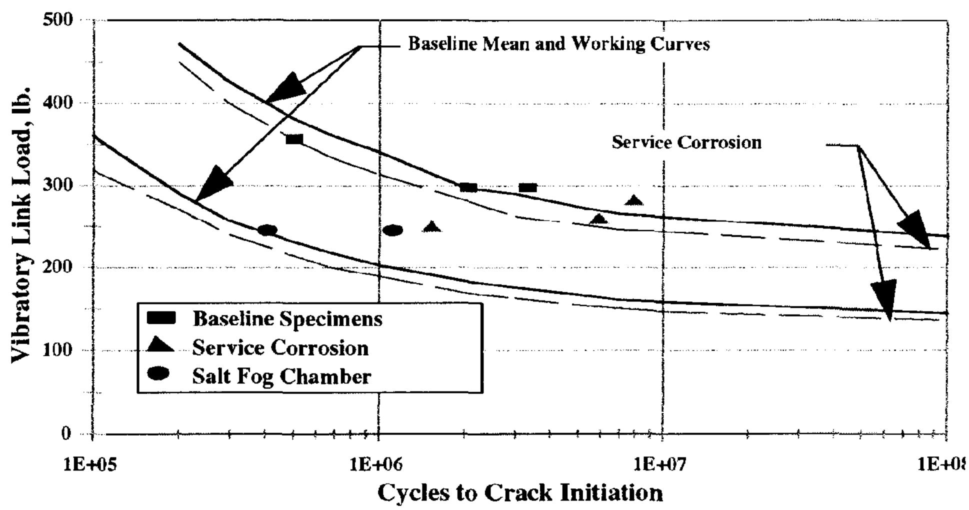


Fig. 9 Effect of corrosion on the S-N curve of the S-76 tail rotor pitch horn (Ref. 16)

A second example was the tail rotor pitch horn of the S-76 helicopter, Ref. 16. This component also experienced corrosion. The pitch horn is made of an aluminum alloy with a 22,000-hour retirement life. A "worst case" condition was used to evaluate failure due to corrosion, Fig. 9. In this case, instead of using inspections to monitor the structural integrity of the pitch horn, its retirement life was reduced to 12,000 hours. This decision was based on the fact that some corrosion had been seen to occur on all pitch horns that had been inspected.

Where corrosion is a problem, extreme vigilance must be exercised to prevent the problem from becoming severe and requiring extensive repairs. Evidence of this severity has been seen in the civilian fixed-wing fleet. This situation is often aggravated when aircraft, subjected to short-term changes of ownership or operators, do not appear to receive adequate maintenance

Multi-Site Damage

Multi-site damage, also called wide-spread fatigue damage, occurs in structures with "similar details operating at similar stresses where structural capability could be affected by the interaction of similar cracks."

In fixed-wing aircraft typical structures where MSD could occur would be wings and empennages with chordwise splices and rib-to-stiffener attachments. This type of damage is also common in fuselages with lap splice joints. Generally in the fuselages of fixed-wing aircraft and helicopters, with all the rivet holes, lugs, and other fittings, MSD is very likely to occur in several locations. MSD is less a safety issue in helicopters than for fixed-wing aircraft because of the pressurized cabins found in the transport civilian fleet. When a series of small cracks link up to form one large crack in the structure of a pressurized cabin, the extra energy released could be a serious safety hazard that would not occur in the non-pressurized fuselages of helicopters. This was the situation in the structural damage suffered by the Aloha aircraft in 1988 where the safety of the aircraft was jeopardized.

It appears that even though MSD has not been a safety issue in helicopters, general cracking is observed more and more. This is one area of concern where regular inspections can be of great value. In Reference 7, Eurocopter estimates that about 20 accidents and major incidents over 43 million flight hours could have been avoided by periodic inspections of the helicopters. If fleet surveys of problems are systematically recorded, trends can be identified and major structural failures can be avoided in the future. If multiple cracking is found early through inspection, modifications can be made and catastrophic conditions can be preempted from developing. The military believes that helicopter airframes are not "tracked" as well as the dynamic components. Since airframes are the structures where MSD is most likely to develop, some type of tracking (inspection) program should be instituted to preclude much larger and expensive solutions later in service.

One example where a cracking problem like MSD has been noted is in the Royal Navy's EH-101 helicopters, Ref. 17. During an extensive fatigue test program cracks were noticed to form in the rear fuselage where conventional skin-stringer designs are used. This part of the airframe was identified as one to be managed on an "on-condition" basis. This is a good example where a thorough initial fatigue test program has identified MSD and developed a good inspection program to extend the safe service life of the EH-101 helicopter.

Programs are now being instituted to monitor areas where future problems may occur and take action early to mitigate them and extend component service lives. A case in point is the U.S. Navy's Helicopter Integrated Diagnostic System (HIDS). One aspect of HIDS is to track vibration in helicopters in order to maintain smooth rotor blade operations thus reducing fatigue-type loads and alleviate cracking problems like MSD.

Loads and Usage Monitoring

Loads and usage monitoring is one of the key technical topics of the day. It makes little difference how exact the life prediction model is if the magnitudes of the loads are not known or are inexact. When two different pilots fly the same maneuver on the same helicopter the measured loads can differ by as much as a factor of two. When a syllabus of typical maneuvers were flown by six pilots on the same helicopter, the average coefficient of variation in measured loads for all maneuvers was between 7 to 10 percentages. Consequently, in order to include "unknown" loads in the loads analysis for calculating the safe retirement life, only the top-of-scatter loads are considered. This assumes that the maximum load measured in a maneuver occurs throughout the maneuver. However, the loads experienced by a structure not only varies from aircraft to aircraft but also on how an aircraft is flown. Thus, part retirement based on average aircraft usage is difficult to quantify. Again, as aircraft age, the operation of individual aircraft are increasingly and substantially different from the average of the whole fleet, and estimates of the retirement lives of components based on average aircraft usage are inadequate. The deterministic loads and allowable used in safe-life methodology has also been attributed for the frequent inspections recommended by manufacturers. A probabilistic approach has been suggested for a more efficiently managed fleet.

As it was previously noted under the section on MSD, a system like the U.S. Navy's HIDS can play a role in loads and usage monitoring. Certainly with the rapid advancements in algorithms, sensors and computer technology smaller and more sophisticated usage and loads monitoring systems are now available, and it is expected that an on-board life prediction system will soon be available to record actual loads for the usage monitoring system. This will account for the variation in loads caused by the pilot's input. In the case of the HIDS system, the use of an automated diagnostic system for helicopters has been shown to provide early warning of damage and wear before the occurrence of failure. This has been demonstrated by the U.S. Navy on the drive train components of the UH-60 helicopter using HIDS,. Ref. 18. The HIDS system is designed to detect early, and monitor the progression of, incipient "fault condition." Thereby, the health of a component can be known at any point in time, the accumulation of damage can be tracked and component replacement at normal overhaul times on aging aircraft can be effectively managed. Thus, HIDS not only maintains the helicopter's structural integrity but also improves availability and reliability while minimizing cost through scheduled maintenance.

The variation in usage from aircraft to aircraft occurs not only with changing requirements of the operating agency but also when the aircraft is operated by several agencies. This is the case of the European multi-nation Tornado aircraft, a total of about 1000 of which have been procured by three nations through 1997. It was originally conceived as a low-level strike and reconnaissance aircraft. With three different nations using this aircraft, the Tornado is now a multi-role

aircraft and loads are accumulated at rates different from the original design spectrum. This example is typical of the dilemma faced by all aircraft manufacturers.

Nondestructive Inspection and Reliability

Perhaps the most important aspect of assuring structural integrity in aging aircraft and yet probably the weakest link is that of identifying and locating damage through the use of nondestructive inspection (NDI) methods. If the service life of an aircraft is extended, damage is inevitable, which must be detected reliably and repaired properly. Often, it is not a matter of how small a damage can be located, but how reliably can such small areas of damage be found. For most metal structures, this damage is either corrosion or a crack. The question in NDI is not how small a crack can be detected, but how large of a crack can be missed from being detected.

In the technology of managing structural life through inspections the U.S. Air Force has the most experience since its formal adoption of the ASIP program when MIL-HDBK-1530, Ref. 9, was published in 1972. In regard to a crack size that can be detected with a high degree of reliability, the Air Force in its damage tolerance design philosophy uses a roque flaw (the largest flaw likely to exist) of 1.27 mm as its standard in fixed-wing aircraft. In regard to how small a crack must be found in helicopters to insure their structural integrity, crack sizes of the order of about 0.4 mm are often quoted. While some sources suggest that eddy current can locate cracks of these sizes fairly reliably, others have found that the smallest crack that can be detected with a high degree of reliability is 0.8 mm. Reliable crack detection is of prime importance as the rotorcraft community attempts to move towards a life management system based on extended safe-life of its structures. In the current environment of mostly safe-life designs, the rotorcraft community is limited in its ability to manage aging helicopter structures because the Palmgren-Miner's rule in safe-life methodology does not physically model the initiation and growth of a crack. Some sources even believe that a 0.4 mm initial crack size is too large for damage-tolerant designs of helicopter components and that crack sizes as small as 0.2 mm must be the design basis. What is important is reliable detection of any crack in the prevailing environment under which the inspection is conducted. As the helicopter community continues to design affordable structures to damage tolerance and to stringent weight requirements, it is obvious that increasingly small cracks are required to be reliably detected. To increase the reliability of detecting cracks, automated inspection systems are required to eliminate the human error in reading and interpreting results. Nondestructive inspection methods and acceptance/failure criteria are described in greater detail below.

The Aging Problem of Military Aircraft

From statements made above and as shown in Table 1, the military aging problem is much more severe than that of the civilian fleet. In the case of military aircraft, it is becoming increasingly difficult to justify the budget to regularly replace the military fleet. The military has also not been able to-date to develop a database of aging aircraft problems in order to identify the time and rationale when structural parts of a helicopter should be replaced.

In the U.S. Army, which has a fleet of about 5,000 helicopters, the Aircraft Condition Evaluation (ACE) program was recently initiated to re-engineer older helicopters to as-new condition. This program has already revealed that while new helicopters have time between overhauls of 1,000 hours or more, older, refurbished parts often have only a few hundred hours between overhaul. It is, therefore, not cheaper to maintain older helicopters than to buy new ones because evidence shows that the cost of maintaining helicopters rises continuously with the age of the helicopter. A case in point is the U.S. Army's CH-47D Chinook helicopters which are re-engineered and refurbished from the CH-47A. The increasing cost for maintaining the CH-47D helicopters over a period of nine years is shown in Fig. 10, Ref. 1. As the hours of operation of a helicopter increases, the military budget is doubly penalized with higher operation and support (O&S) cost and higher maintenance cost because the helicopter is further aged. The O&S cost for CH-47D helicopters in terms of flight hours is shown in Fig. 11, Ref. 1.

The U.S. Army continues to maintain its helicopter fleet with a safe-life design philosophy. If cracks occur, the problems are often managed using a damage-tolerant inspection procedure. However, if the component that is experiencing these problems is redesigned, the service life is again calculated on the basis of the safe-life approach. The Army is considering adopting more and more damage tolerance types of procedures, but the cost and reliability of NDI remain the principle barrier for wider use of the damage tolerance methodology. The remoteness of some deployment sites and the use of the foot soldier for inspections are two reasons why the Army considers the damage tolerance approach to be too risky at present.

The U.S. Navy also has an aging fleet of helicopters, with the average age of about 19 years, Fig. 1. The Navy expects to alleviate the aging issue through an aggressive replacement program starting in 2005; however, the rate of acquisition will depend on the available budget. The Navy's procedures for managing their fleet are almost the same as the U.S. Army's structural life management philosophy. That is, the safe-life methodology sets retirement times. In case of failure, damage tolerance principles are used to address problems in the field while a safe-life redesign is undertaken to increase the service life of the component, Ref. 3. The U.S. Navy is not too encouraged with the damage tolerance approach because

of the difficult environment in which it operates, and because shipboard inspection is often undertaken by seamen without extensive experience, Ref. 2.

The U.S. Air Force, while desiring to move towards a damage tolerance philosophy for helicopters because of the successful application of its ASIP program to its fixed-wing fleet, has depended on the Army for its structural integrity methods because its fleet of helicopters is small.

NDI AND ACCEPTANCE/FAILURE CRITERIA

Overview

It has been discussed above why application of damage tolerance principles are being inhibited by the cost and reliability of NDI methods. This section describes the NDI methods for composites structures; the same principles are generally applicable to metallic components. Composite structures are more difficult to inspect because they are made of non-homogeneous materials and are manufactured by a variety of processes each with its specific requirements for quality. In addition, special attention must be given to composite structures in order that internal defects or damage can be detected and assessed. In metallic materials, flaws are modeled and linear elastic fracture mechanics applied to predict failure. Unfortunately a unified failure model is still under development for fiber-reinforced composite materials. Experimental methods for detecting internal flaws in composite structures and comparing the flaws to reference standards are the only means for evaluating the structural integrity and the residual level of performance in safety-critical applications.

Non-destructive inspection methods identify manufacturing and in-service defects in structures without degrading their quality or affecting their serviceability. Defects can be external and internal to the structure. External defects can be visually inspected, such as dimensions, finish, and warpage. Internal defects

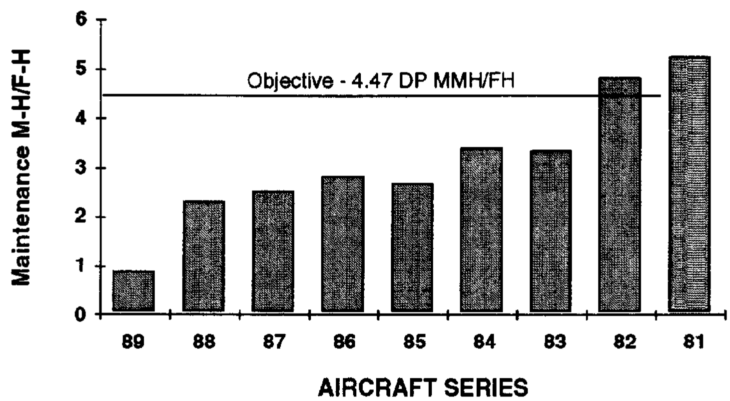


Fig. 10 Maintenance man-hours per flight hour for the Chinook helicopters in the U.S. Army's inventory (Ref. 1)

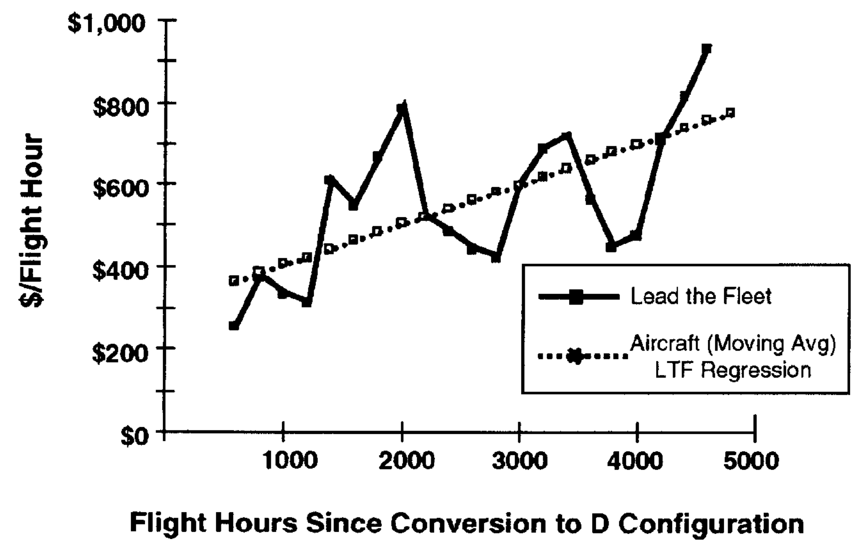


Fig. 11 The O&S cost of Chinook helicopters with respect to hours of operation (Ref. 1)

of most concern in composites are delaminations, inclusions, voids, resin-rich and resin-starved areas, fiber misalignments and breakage, and debonds. NDI is part of quality assurance that controls the manufacturing process in order to meet design specifications, produce repeatable products and reduce cost.

Composite materials are made of two constituents: fibers for strength and a matrix to bind the fibers to shape. Defects will inevitably occur in composite structures. NDI is used to evaluate the criticality of the defect(s), i.e., number, size(s) and location(s). The implementation of NDI must, therefore, begin during design by arranging structural details to facilitate inspection, and with analyses to qualify the arrangements and identify the critical sizes and locations of defects in terms of the capabilities of the available techniques. In order to conduct NDI efficiently and accurately, reference standards must be established which take into account the design, equipment, the minimum defect size that can be detected, types of defects and acceptance criteria. The acceptance criteria are developed through destructive tests and analyses to quantify the criticality of defects in a flawed structure.

Types of Internal Defects

Internal defects typically occur due to variations in the material from batch to batch and due to the variables in the manufacturing process. Most defects can be easily prevented through proper process control, of which NDI is an integral part. The internal defects of most concern for manufacturers of composite structures, and where NDI alone can assess the "damage" caused by these defects in order to assess the residual level of performance are described in Table 3 (Ref. 12).

NDI Methods

The integrity of composite structures is evaluated by identifying (sizes and locations of) defects using NDI methods. Several NDI methods are in use, but none of them quantifies the integrity of the structure. The results of NDI are compared with drawing specifications and reference standards in order to identify the sizes and locations of defects. A structural analysis and test program then quantify the integrity of the structure. Descriptions of NDI methods are available in several publications, including Ref. 19-25. The advantages and disadvantages of the more widely used methods are shown in Table 4 (Ref. 12), and their efficacy in detecting defects in Table 5 (Ref. 12). The most common methods for NDI are visual, radiographic and ultrasonic inspections. Specialized inspection methods are used for specific applications. For example, Cat-Scan has been used to inspect alignment and compaction of fibers.

Acceptance Criteria for Production Parts

Since a composite structure is a complex assembly of elements made from several material types and forms, it is inevitable that defects of various shapes and sizes will be present in production parts. During the design process the criticality of defects are analytically determined, which then establishes the acceptance criteria. The acceptance criteria vary widely with the type of defect, the structure being inspected and the NDI method used. For example,

1) the acceptance criteria are more stringent for primary structures than for secondary structures,
2) the acceptance criteria for voids and porosity can be given in terms of signal attenuation for ultrasonic inspection or in terms of linear dimension and area when radiography is used, and
3) the limiting area of acceptability can be for individual voids or for an envelope around several scattered defects.

Non-destructive inspection is the joint responsibility of engineering, manufacturing and process control to assure that composite structures are manufactured to a quality consistent with the design requirements. To assure early detection of defects, specifications of acceptance criteria for each NDI method are documented on the basis of the minimum defect size that can be detected. However, these documents are general in their requirements, and specific criteria for primary components or at specific locations are included in engineering drawings. Specific locations may be critical because they are in highly stressed areas or because they are difficult to inspect.

The documents on general acceptance criteria include the acceptance and repairable limits and the repair procedure. If a defect can be repaired, the component is directed to process specifications and then re-evaluated through NDI. Typical acceptance criteria for laminated structures are given in Table 6 (Ref. 12). Since acceptance criteria are based on manufacturing experience and are peculiar to the NDI equipment in use, the list of defects given in the tables will vary and their limits may be refined to the unique capabilities of each manufacturer.

The acceptance criteria for composite sandwich structures include both typical defects as well as defects that are unique to the designs of a manufacturer. The design details are especially important for composite components of the rotor system, e.g., the blades. Since the blade is constructed to meet the specifications of static and fatigue strengths, stiffness, and the distribution of weight, specific details of blade elements - the spar, leading edge, trailing edge and after-body - may be separately identified in the acceptance criteria. The number of details identified depends on the manufacturing method used for the blade elements and the assembly procedure adopted. Typical acceptance criteria for defects in sandwich structures are given in Table 7, Ref. 12.

Failure Criteria for Development Test

Composites under inspection will always indicate structural defects either visually or by NDI. These defects have to be identified as structurally significant before further substantiation is undertaken. As stated previously, composites are inherently flawed, with some level of each of these flaws acceptable at any given location of the component. By "acceptable," it is meant that the structural integrity of the component is not compromised by the presence of the flaw.

The declaration of a structural failure then must be postponed until it is verified that the observed defect can grow to be unstable to the point where structural integrity is affected. In safe-life testing, it may be appropriate to consider the initiation point as the number of test cycles where the defect was first observed or detected. If the defect growth is arrested and component structural integrity is not reduced, the defect can be declared as "cosmetic" and ignored in analysis and in service.

In extended safe-life testing, the initial detectability must be defined in visual terms or in terms of one of the specific methods in Table 4, which then becomes the standard method for inspection in service. The rate of growth of the defect is then monitored in the test, and the equivalent flight hours to unstable crack length is used to calculate the in-service inspection intervals.

Table 3. Descriptions of Defects in Composite Structures (Ref. 12)

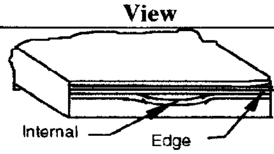
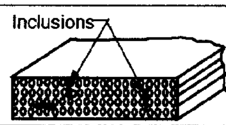
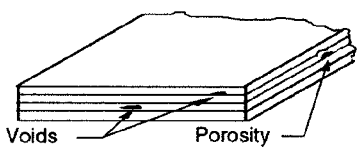
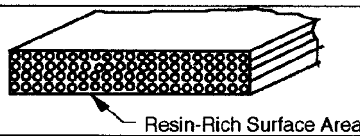
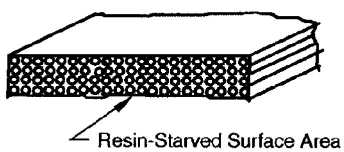
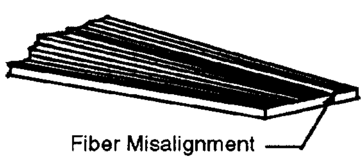
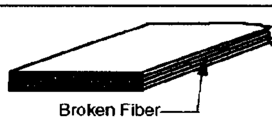
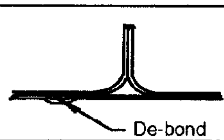
Defect	View	Description
Delamination		Delaminations are separations within plies of a laminate, and caused by improper surface preparation, contamination and embedded foreign matter.
Inclusions		Inclusions are foreign matter embedded in and between laminae.
Voids and Porosity		Voids and porosity are entrapped air and gas bubbles, and are caused by volatile substances, improper flow of resin and unequal pressure distribution. Voids are clustered in the resin, while porosity are pockets within the solid material..
Resin-Rich Area		Resin-rich areas are localized, and filled with resin or lacking in fiber. This defect is caused by improper compaction or bleeding.
Resin-Starved Area		Resin-starved areas are localized with insufficient resin evident as dry spots, or having low gloss or where fibers are exposed. This defect is caused by improper compaction or bleeding.
Fiber Misalignment, Wrinkling, Buckling		Fiber misalignment is a distortion of the plies resulting in changes from the desired orientation, or in fiber wrinkling and buckling. These defects are due to improper lay-up and cure.
Fiber Breakage		Broken fibers are discontinuous or misplaced fibers due to improper handling or lay-up.
De-bond		De-bonds occur between different details of the built-up structure. Lack of bonding is due to contamination of the surface, excessive pressure or bad fit.

Table 4. Efficiencies of NDI Methods for Composites (Ref. 12)

NDI Method	Application	Advantages	Disadvantages
Visual	1. Surface defects	1. Simple 2. Economical	1. Limited information 2. Surface defects, finish
Coin Tap	1. Large voids, debonds and delaminations	1. Simple 2. Economical	1. Limited information 2. Difficult with very large structures.
Radiographic	1. Internal defects in sandwich structures 2. Edge delamination and damage	1. Easy to use 2.Equipment readily available 3. Extensive information on the state of damage 4. Permanent record	1. Radiation safety concerns. 2. Relatively expensive 3. Penetrant required 4. All damages (thin flaws perpendicular to the beam) may not be detected.
Ultrasonic (C-Scan)	1. Voids and porosity 2. Delaminations	1. Very accurate results 2. Relatively low-cost process 3. Permanent record 4. Thick structures 5. Can be automated	1. Slow operation 2. Couplant medium needed 3. Bulky, relatively expensive equipment 4. Difficulty with complex geometry (core, etc.)
Acoustic Ultrasonic	1. Delaminations and voids 2. Fiber breakage 3. Fiber-matrix interface 4. Fiber orientation	1. Higher resolution than C-scan. 2. Quickly evaluates for acceptance or rejection 3. Permanent record.	1. Limited to small areas 2. Dependent on surface geometry since the medium is the structure
Thermographic	1. Delaminations and voids 2. Debonds	1. Simple system 2. Quantitative results 3. Real-time images 4. Surface contact not required.	1. Experience required to size and type of defect 2. Limited size of structure 3. Thin specimens only
Acoustic Emission	1. All defects	1. Continuous monitoring 2. Global	1. Structure is under load 2. Voluminous data 3. Simple geometry
Holographic	1. Delaminations and voids 2. Core-to-skin de-bonds	1. Rapid evaluation 2. Relatively inexpensive 3. Surface preparation not required 4. Permanent record 5. Real-time image	1. Sensitive to vibration if not coupled 2. Optically accurate alignment required 3. Laser safety concerns
Shearographic	1. Delaminations and voids 2. Impact damage 3. Cracks in holes 4. De-bonds	1. Rapid evaluation 2. Relatively inexpensive 3. Portable equipment 4. On-aircraft inspection 5. Permanent record 6. Real-time image	1. Laser safety concerns
Eddy Current	1. Surface defects	1. Relatively low-cost 2. Can be automated	1. Limited to good electrical conductors
Edge Replication	1. Surface defects 2. Initiation and progression of cracks	1. Ply-by-ply record 2. Saturation number of transverse cracks 3. Simple, rapid 4. Permanent record	1. Surface defects only 2. Often limited to coupon-type specimens

Table 5 Capabilities of NDI Methods for Composites (Ref. 12)

NDI Method	Core Damage	Delamination	De-bond	Fiber Break	Fiber Misalignment	Impact Damage	Inclusion	Resin Variation	Void
Visual		S	S	S	S	S		S	S
Radiography	A	C	C	C	B	B	A	A	A
Ultrasonic	B	A	A	B	B	A	A	A	A
Acoustic Ultrasonic	B	A	A	A	B	A	A	A	A
Thermo-graphic	C	B	B			B	C	C	B
Acoustic Emission		A	A	A		A			
Holographic	A	A	A	B		A	B		A
Shearo-graphic	A	A	A	B		A	C		A
Eddy Current[a]	B	B	B	A		B	B	B	A

Legend - A: Good detection; B: May not detect minor damage; C: Detection when defect is large; S: Detection at surface or edge

Notes: [a]: For good electrical conductors

Table 6 Typical Acceptance Criteria For Laminated Composite Structures (Ref. 12)

No.	Discrepancy	Acceptance Limit[a] (in.)	Repairable Limit[a] (in.)	Repair Procedure[b]
1	Surface Depressions	0.25 dia., 0.03 deep or less than 25% of laminate thickness.	>0.25 dia, <0.05 deep and no fiber damage.	Sand, clean, fill with epoxy, cure, sand to dimensions and verify.
2	Surface Pin Holes	0.10 dia. or 0.03 dia. holes over 10% of laminate area.	<0.25 dia. or >0.03 dia. covering 70% of laminate area.	Sand, clean, fill with epoxy, cure, sand to dimensions and verify.
3	Surface Cracks	None.	Not applicable.	Not applicable.
4	Resin-Rich	0.03 thick.	None.	Sand (avoid fiber damage), clean, verify dimensions.
5	Resin-Starved	All, if only on surface ply.	None.	Sand (avoid fiber damage), clean, brush epoxy, cure, sand to dimensions, verify.
6	Frays, Burrs	0.13 at machined edge.	>0.13 or affecting assembly.	Trim, apply adhesive, cure, sand to dimension, verify.
7	Surface Inclusions	0.10 sq.in. each with 5% of laminate area.	Greater than acceptable limit.	Sand, clean, fill with epoxy, cure, sand to dimension, verify.
8	Warps	0.01 gap from flat surface with 10 LB applied every 12 inches.	None.	Not Applicable.
9	Delaminations, Radii Bridging	0.125 sq.in., and away from other indications[c].	(Dimension varies with distance from edge.)	Drill 0.0625 or smaller holes, inject epoxy, clean, cure, sand to dimension, verify.
10	Porosity, Voids	3%-10% depends on class of structure[d].	None.	Not applicable.

Notes:
[a] The limits shown vary with manufacturers and can be overridden by specifications on engineering drawings.
[b] The repairs are conducted to process specifications. Only the outlines are presented here.
[c] Several limits might be included to reflect importance of structure (primary, secondary, redundant, etc.) or NDI method used. Limits can also be given in terms of attenuation of the signal.
[d] The limits vary with the importance of the structure and the requirements of the certifying agency. Specific limits are often stated on engineering drawings.

Table 7 Typical Acceptance Criteria for Honeycomb Sandwich Composite Structures (Ref. 12)

No.	Discrepancy	Acceptance Limit[a] (in.)	Repairable Limit[a]	Repair Procedure[b]
1	Core Distortion	Must be correctly in place in lay-up tool when tacked.	Unlimited for Nomex cores None for fiberglass cores.	Re-form Nomex cores and assemble. Not applicable for fiberglass cores.
2	Core Crushing	(Depends on location. Usually) 1 cell deep, 1.00 in any direction.	None.	Remove and replace.
3	Core Buckling	1.00 in any direction and 1% buckled of total depth, with next buckled area at least 6.00 away.	None.	Remove and replace.
4	Core Nesting	(Number of cell rows depend on particular location.)	None.	Remove and replace.
5	Bond Line Thickness	1 layer: 0.003 to 0.015 2 layers: max. 0.020, etc.	None.	Not applicable.
6	De-bonds	Must be continuous.	Unlimited only where repairable; otherwise none.	Repair where possible; otherwise cut out, replace, re-bond.
7	Foam Adhesive	Generally a gap length of 3-4 cells over 6.00; but depends on particular location.	Unlimited only where repairable; otherwise none.	Repair where possible; otherwise cut out, replace, re-bond.
8	Discontinuous Bond Line Transverse to Splice	0.1 visible width.	None.	Not applicable.
9	Foreign Material in Bond Line	None.	None.	Not Applicable.

Notes: [a] The limits shown vary with manufacturers and can be overridden by specifications on engineering drawings.
[b] The repairs are conducted to process specifications. Only the outlines are presented here.

SUMMARY AND CONCLUSIONS

The helicopter industry has been aware for sometime of the importance of addressing the issues of aging aircraft. The methods for substantiating the structural integrity of components leads to possible approaches for addressing problems of aging aircraft. These methodologies for assuring the structural integrity of structures have been discussed in detail. With the multiplicity of materials, manufacturing processes and operating spectra, the extended safe-life approach appears best to address issues of both initial qualification and aging aircraft. The extended safe-life approach though appropriate for civilian operations is not fully favored by the military. The military prefers the safe-life approach for logistical reasons and because of difficulties in using NDI methods under operational conditions.

A discussion of the available NDI methods has been provided in terms of acceptance criteria to help identify problems associated with NDI and to provide directions for alleviating the difficulties encountered by the military. The severe conditions under which the U.S. Navy has to operate is shown in Fig.12, and some times under these conditions, damage tolerance evaluations are conducted by seamen, the average age of whom in the current U.S. Navy is 19 years, Ref. 2. Similarly, the evaluations by the foot soldier in the U.S. Army may be conducted under less than optimum, though a different set of difficult, conditions.

The large number of accidents caused by operational and environmental conditions beg further attention because these exceed accidents due to structural failures by several magnitudes. To reduce these accidents, further

Fig. 12 Operating conditions of the U.S. Navy (Ref. 3)

development of all-weather helicopters are required, so that helicopters are equipped with such features as fly-by-wire and health and usage monitoring systems.

The technology for analysis exists today to assure structural integrity issues of aging helicopters. The extended safe-life approach encompasses the best of several methodologies to make the qualification of structural integrity affordable. In conjunction with qualification assurance, the development and validation of simple-to-use health, structural and usage monitoring systems will further improve the integrity of structures to meet the increasingly stringent requirements of both the civilian and military operators, and will also reduce accidents due to non-structural causes.

Another aspect that has been highlighted is the need for a database which should be a systematic record of the failures experienced and of inspection results of the causes of degradation and failures. This database will provide validation that manufacturers require for further improving the structural integrity and for providing optimum guidance to helicopter operators. With innovative designs, advanced manufacturing processes, improved NDI techniques and a systematic database on experiences encountered, quality assurance for helicopters can eliminate structural failures in worldwide, all-missions operations.

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